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## RESEARCH MEMORANDUM

PRELIMINARY INVESTIGATION OF AIRFOIL CHARACTERISTICS

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IN THE Langley ANNULAR TRANSONIC TUNNEL

By Louis W. Habel and James H. Henderson

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## RESEARCH MEMORANDUM

## PRELIMINARY INVESTIGATION OF AIRFOIL CHARACTERISTICS

## IN THE LANGLEY ANNULAR TRANSONIC TUNNEL

By Louis W. Habel and James H. Henderson

## SUMMARY

In order to reduce the spanwise angle-of-attack variation due to tunnel-wall boundary layer in the Langley annular transonic tunnel, the length of the annular path ahead of the test section was considerably reduced. With this shortened entrance an experimental investigation was made of an NACA 66-006, an NACA 65-110, and two symmetrical double-wedge airfoils in the Mach number range from approximately 0.6 to slightly over 1.0 at angles of attack from  $0^\circ$  to  $4^\circ$  in an attempt to evaluate the validity of data obtained with the new tunnel configuration.

Although the lift-curve slopes measured in the Langley annular transonic tunnel at subsonic Mach numbers have been substantially increased by reducing the tunnel entrance length, the measured lift-curve slopes remain lower than would be expected from extrapolations of low-speed data for corresponding airfoils obtained in the Langley two-dimensional low-turbulence tunnels.

The pressure distributions measured in the Langley annular transonic tunnel with the shortened entrance are in agreement with data from other sources and with theory in most respects if the comparisons are made at equal values of normal-force coefficient, although the measured pressures appear to be slightly more positive than those with which they are compared.

## INTRODUCTION

The Langley annular transonic tunnel is being developed by the NACA to obtain two-dimensional pressure-distribution data in the transonic speed range. A description of the tunnel and a discussion of some of its limitations are presented in reference 1. It was shown that a serious spanwise angle-of-attack variation due to the boundary-layer

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thickness of the axial flow existed for models tested in the tunnel even though the boundary-layer thickness of the axial flow was controlled to some degree.

Part of a systematic program evolved to investigate the limitations of the equipment and to determine the sources of possible error in the data included a plan to reduce the spanwise angle-of-attack variation from that indicated in reference 1. In accordance with this plan, the length of the annular axial-flow path ahead of the rotor was substantially reduced in an effort to reduce the undesirably large boundary-layer thickness due to the axial flow.

The purpose of this paper is to present some of the results obtained in the Langley annular transonic tunnel with use of wall-boundary-layer control after the entrance was shortened and to compare these results with data from other sources and with theoretical calculations.

#### APPARATUS

The Langley annular transonic tunnel is shown schematically in figure 1. Two concentric circular cylinders are arranged with a 3-inch annulus between them. A test airfoil equipped with pressure orifices at the midspan station is attached to a rotor equal in diameter to the inner cylinder and may be rotated in the annulus at any speed up to low supersonic velocities. The diameter of the path on which the model midspan station rotates is 5 feet. A small axial velocity is induced through the annulus by a fan. Thus, the helix angle of the flow and therefore the angle of attack of the test airfoil may be controlled. The test airfoils very nearly span the 3-inch annulus and have 4-inch chords along their entire span. At a Mach number of 1.0 the Reynolds number of the model is approximately 2.3 million.

Recently, in an attempt to reduce the wall-boundary-layer thickness due to the axial flow, the entrance section to the Langley annular transonic tunnel was considerably shortened by installing a rounded entrance on the outer shell approximately 22 inches ahead of the test section. The change which was made is indicated in figure 1. The original entrance is shown as a dashed line.

The boundary-layer removal slot approximately 12 inches ahead of the test section was employed with the shortened section to reduce the boundary-layer thickness to the minimum value obtainable with the present system. In figure 2, the spanwise angle-of-attack variation is compared for the two entrance configurations. The curves presented were computed from the results of axial-velocity surveys across the annulus and are for an angle of attack of  $0^\circ$  at the midspan station of the

airfoil. For other angles of attack the curves would be shifted slightly. For example, when the midspan station of the airfoil is operated at an angle of attack of  $4^{\circ}$ , the 5-percent-span station operates at an angle of attack of  $3.4^{\circ}$ . Note that when the entrance was shortened, the greatest reduction in angle-of-attack variation occurred at the model tip; thus additional improvement could probably be obtained at the model root by installing a rounded entrance on the inner cylinder near the test section.

### TESTS

As data obtained in the Langley annular transonic tunnel are subject to question because of the unconventional design of the apparatus producing the data, comparisons of data obtained from this tunnel with data from other sources and with theory are necessary. Consequently, most of the tests in the Langley annular transonic tunnel to date have been made to evaluate the accuracy of the data obtained from the tunnel rather than to produce a quantity of pressure-distribution data.

In the present investigation, a number of airfoils were tested over a Mach number range from approximately 0.62 to slightly over 1.0 at angles of attack from  $0^{\circ}$  to approximately  $4^{\circ}$ . The airfoils which were tested were the NACA 66-006, the NACA 65-110, and two symmetrical double-wedge airfoils which differed only in that one was 6 percent thick and one was 10 percent thick.

The NACA 66-006 airfoil was tested primarily to evaluate the effect on the data of the change made in the tunnel configuration since this airfoil was used in the investigation reported in reference 1. The NACA 65-110 airfoil was tested to obtain pressure distributions which could be compared with pressure distributions measured in flight for an NACA 65-110 airfoil. The two double-wedge airfoils were tested because the theoretical pressure distributions were available for these airfoils at a Mach number of 1.0 for the nonlifting condition in two-dimensional flow.

### DISCUSSION

The data obtained for the various airfoil sections are presented and compared with other experimental data and with theoretical calculations in figures 3 to 10. Some force and moment data obtained from integration of the pressure-distribution diagrams are presented in figures 11 and 12 to indicate the scope of the data which can be obtained from the Langley annular transonic tunnel.

In figure 3 the lift-curve slopes, or, more accurately, the normal-force-curve slopes, are presented for an NACA 66-006 airfoil at a Mach number of 0.625 for the two entrance conditions previously described and shown in figure 1. For comparison, the theoretical low-speed lift-curve slope of  $2\pi$  per radian is shown extrapolated to a Mach number of 0.625 by the Glauert-Prandtl method (reference 2). The lift-curve slope measured for an NACA 66-006 airfoil in the Langley two-dimensional tunnel at low speed (reference 3) is within a few percent of  $2\pi$  per radian. Although shortening the entrance caused a marked increase in lift-curve slope of the NACA 66-006 airfoil as measured in the Langley annular transonic tunnel, the measured lift-curve slope remains considerably lower than the theoretical value. It is of interest, however, that the zero-lift condition for a symmetrical airfoil occurs at practically zero angle of attack for tests made with the short entrance; this result indicates that data obtained for the nonlifting condition may be correct.

Some further insight into the reliability of the data obtained from the Langley annular transonic tunnel may be gained from an examination of the pressure-distribution diagrams. The pressure distribution measured at a Mach number of 0.735 and an angle of attack of  $0^\circ$  for an NACA 66-006 airfoil in the Langley annular transonic tunnel is compared with the theoretical pressure distribution for similar conditions in figure 4. The theoretical pressure distribution was obtained from reference 3 and was extrapolated to a Mach number of 0.735 by the Von Kármán-Tsien method (reference 4). Although the measured pressures are slightly more positive than those predicted by theory, the agreement between theory and experiment shown in figure 4 is probably as good as might be expected since discrepancies of the order of those in figure 4 have been found in comparing experimental pressure distributions obtained in the Ames 1 by  $3\frac{1}{2}$ -foot high-speed tunnel and the Langley rectangular high-speed tunnel with theoretical pressure distributions.

The measured pressure distribution is compared with the theoretical pressure distribution for an angle of attack of  $2.2^\circ$  and a Mach number of 0.694 in figure 5. The low-speed lift coefficient for which the theoretical pressure distribution was computed was determined by using the theoretical value of low-speed lift-curve slope;  $2\pi$  per radian. Although the shapes of the curves shown in figure 5 are in agreement, the theoretical curve represents considerably more lift than indicated by the experimental pressure distribution as would be expected from an examination of figure 3.

Shown in figure 6 is a comparison of the pressure distribution measured in the Langley annular transonic tunnel at a Mach number of 0.694 and an angle of attack of  $2.2^\circ$  with the theoretical pressure distribution computed for the same lift as indicated by the measured

curves. The load distributions of the two curves are in good agreement, but, as in figure 4, the measured pressures are slightly more positive than those indicated by the theory.

In figure 7 the pressure distribution measured in the Langley annular transonic tunnel for an NACA 65-110 airfoil section at a Mach number of 0.79 and a normal-force coefficient of 0.386 is compared with the pressure distributions obtained from unpublished flight data for the same airfoil section near the midsemispan station of a wing where fuselage and tip effects would be expected to be a minimum. The Mach number and the normal-force coefficient of the flight data are the same as for the Langley annular transonic tunnel test. It is of interest that although somewhat more positive pressures were measured in the Langley annular transonic tunnel over both surfaces of the airfoil than were measured in flight, the important parameters such as load distribution and shock location are in good agreement for the two tests. The result from the Langley annular transonic tunnel appears to indicate a less sharply defined shock than that measured in flight. The limited number of pressure-measuring orifices in the Langley annular transonic tunnel model prevents a more detailed study of this effect at present. The angle of attack for the flight test is not known, so that an angle-of-attack comparison cannot be made.

Although the comparative data of figures 5 to 7 are of some interest, they are for relatively low Mach numbers. The speed range in the vicinity of Mach number 1.0 is of the greatest interest. In figure 8 the pressure distribution measured in the Langley annular transonic tunnel for the NACA 65-110 airfoil at a Mach number of 1.0 and a normal-force coefficient of 0.409 is compared with a pressure distribution measured in flight for the same conditions. The pressure distribution measured at a Mach number of 1.0 in flight is for the section near the midsemispan station of the wing for which subsonic data are presented in figure 7. As in the case of the subsonic pressure-distribution comparisons, the pressures measured in the Langley annular transonic tunnel are more positive than those to which they are being compared, but the shapes of the curves and therefore the load distributions are in good agreement.

The theoretical pressure distribution for the nonlifting condition at a Mach number of 1.0 has been computed by Guderley for a symmetrical airfoil with a cusped leading edge in two-dimensional flow (reference 5). This particular airfoil section was designed expressly for ease in computing the flow. Recently Guderley in conjunction with Yoshihara extended his original results to include a symmetrical double-wedge or diamond-shaped airfoil of 10-percent thickness (reference 6). The theory was developed for vanishingly thin sections and apparently contains as an inherent part the transonic similarity law as a means for obtaining pressure distributions on airfoils of finite thickness.

In figure 9 the pressure distribution for a 10-percent-thick symmetrical double-wedge airfoil at a Mach number of 1.0 and an angle of attack of  $0^{\circ}$  as measured in the Langley annular transonic tunnel is compared with the theoretical pressure distribution presented in reference 6. In figure 10 a similar comparison is made for a 6-percent-thick symmetrical double-wedge airfoil. The theoretical pressure distribution shown in figure 10 was obtained by extrapolating the theoretical pressure distribution presented in reference 6 for a 10-percent profile thickness to a 6-percent profile thickness with the transonic similarity rule (reference 7).

Because of the nature of the theory of reference 6, results predicted by the theory for profiles of large thickness would be expected to be in error. It is of interest to note, however, that agreement of experimental data for the 6- and 10-percent-thick airfoils with theory is very good. During tests of the airfoils, a boundary layer which is not considered in the theory exists on the airfoils. Also, the corners of the airfoil were slightly rounded rather than extremely sharp. A reduction in boundary-layer thickness or an increase in the sharpness of the airfoil corners might perhaps bring the experimental result into closer agreement with the theory. It is, of course, obvious that the comparisons presented in figures 9 and 10 do not constitute an entirely satisfactory proof that data obtained for nonlifting conditions in the annular transonic tunnel near Mach number 1.0 are completely reliable, inasmuch as the validity of the theoretical pressure distributions may be open to some question.

The pressure-distribution data presented in the preceding figures indicate that data from the Langley annular transonic tunnel are in relatively good agreement with data from other sources and with theory when comparisons are made for nonlifting conditions and for equal normal-force coefficients rather than for equal angles of attack. It is thus believed that the largest source of error in the data from the Langley annular transonic tunnel is an error in the angle of attack. In view of the large increase in lift-curve slope gained by reducing the wall-boundary-layer thickness, one might expect that further reduction of the axial boundary-layer thickness would result in further increases of the lift-curve slope. Some additional investigation, however, has indicated that such is probably not the case.

As the angle of attack is determined directly from the helix angle which, in turn, is a function of the measured rotational and axial velocities, one might suspect that the measured velocities may be in error. The rotational velocity is measured by comparing the frequency output of a small alternator driven by the rotor shaft with known frequencies and is believed to be in error by considerably less than 1 percent. The axial velocity is measured with a conventional pitot-static tube located in the annulus slightly upstream of the test section. Although the pitot-static tube indicates the average axial velocity in

the annulus, the possibility exists that the average axial velocity in the annulus is not the axial velocity from which the helix angle should be computed. It is believed that there may be an induced angle associated with the nonuniform spanwise lift distribution caused by the spanwise Mach number gradient and other effects which are unknown. This phenomenon is being investigated.

As an indication of the type of data obtained in the Langley annular transonic tunnel, the normal-force coefficients of the NACA 66-006 airfoil are presented as a function of Mach number for several values of angle of attack in figure 11. The data obtained for the lifting conditions are characteristic of normal force versus Mach number curves in that, as the Mach number is increased, the normal-force coefficient increases to a peak, then decreases rapidly, and levels off or recovers near a Mach number of 1.0. The angles of attack indicated in the figure are probably in error; for example, at Mach numbers below the critical speed of the airfoil, it is believed that the curve obtained for an indicated angle of attack of  $4^{\circ}$  would be obtained at a somewhat lower angle of attack if the correct value of lift-curve slope were obtained. (See fig. 3.)

Also shown in figure 11 are curves of the experimental and theoretical critical Mach numbers. Note that for the lifting conditions the theoretical critical Mach numbers are lower than those found experimentally, primarily because the pressure at the leading edge of the upper surface of the airfoil as predicted by the theoretical pressure distributions were more negative than those obtained by experiment. However, the pressure-measuring orifice which was closest to the leading edge of the airfoil in the upper surface was located at the  $2\frac{1}{2}$ -percent chord station. It is believed that, if pressure-measuring orifices had been installed nearer the leading edge of the airfoil than the  $2\frac{1}{2}$ -percent chord station, the experimental and theoretical critical Mach numbers would have been in better agreement.

In figure 12 the pitching-moment coefficients about the quarter-chord position are presented for the NACA 66-006 airfoil as a function of Mach number for several values of angle of attack. The moment coefficients generally remain near zero but diverge to negative values as the angle of attack and the Mach number are increased. The angles of attack indicated in this figure are, as in figure 11, believed to be in error.

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## CONCLUSIONS

Although the lift-curve slopes measured in the Langley annular transonic tunnel at subsonic Mach numbers have been substantially increased by reducing the tunnel entrance length, the measured lift-curve slopes remain lower than would be expected from extrapolations of low-speed data for corresponding airfoils from the Langley two-dimensional low-turbulence tunnels.

The pressure distributions measured in this investigation are in agreement, in most respects, with data from other sources and with theory if the comparisons are made at equal values of normal-force coefficient, although the measured pressures appear to be slightly more positive than those with which they are compared.

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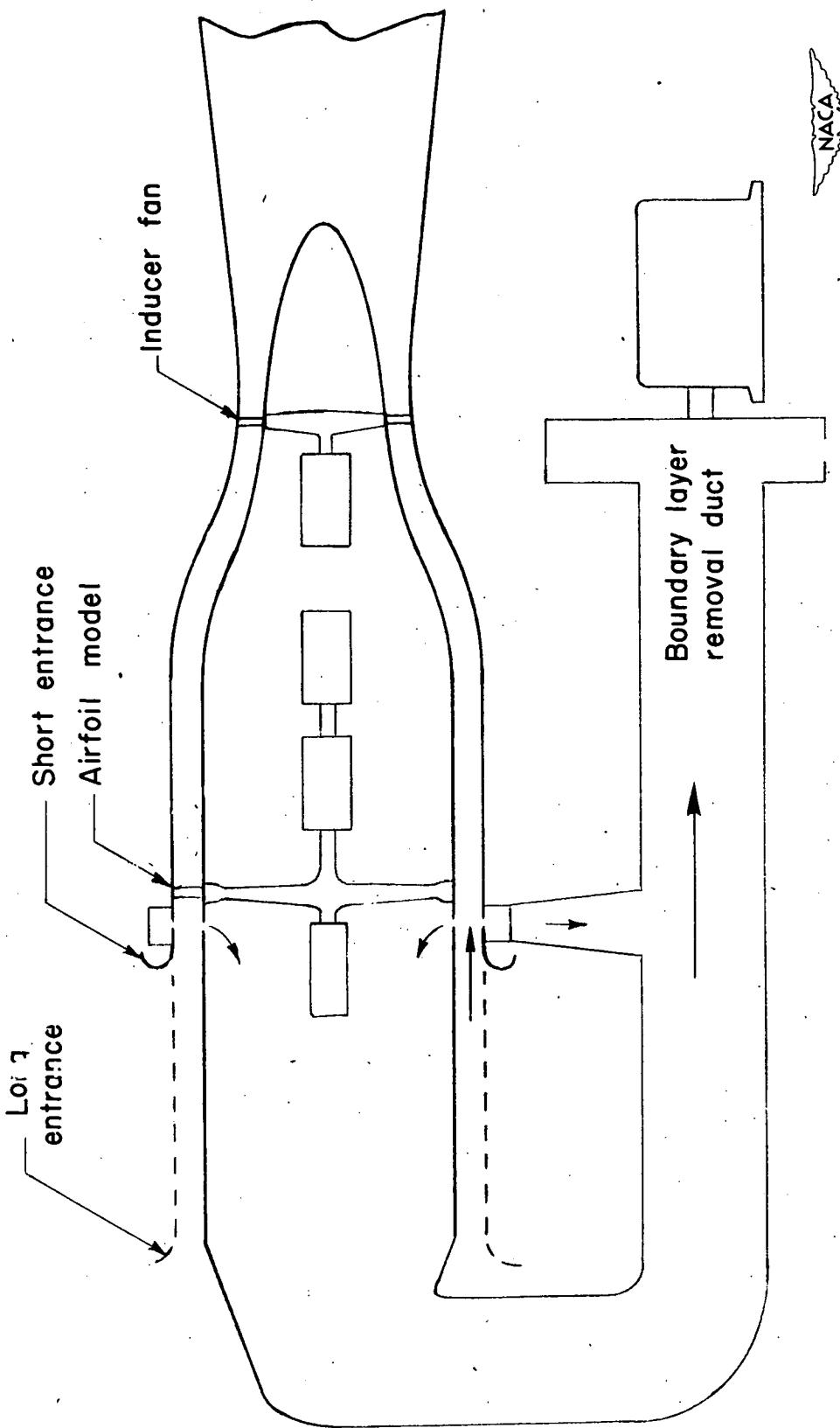


Figure 1.- Schematic diagram of Langley annular transonic tunnel.

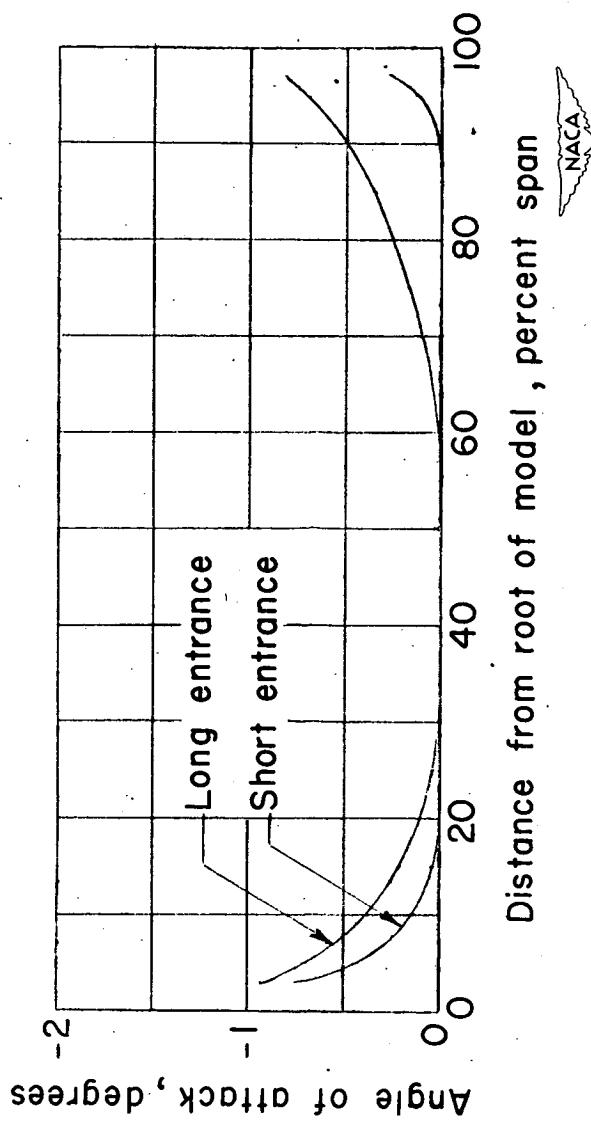


Figure 2.- Spanwise angle-of-attack variation of Langley annular transonic tunnel models due to boundary layer of the axial flow.

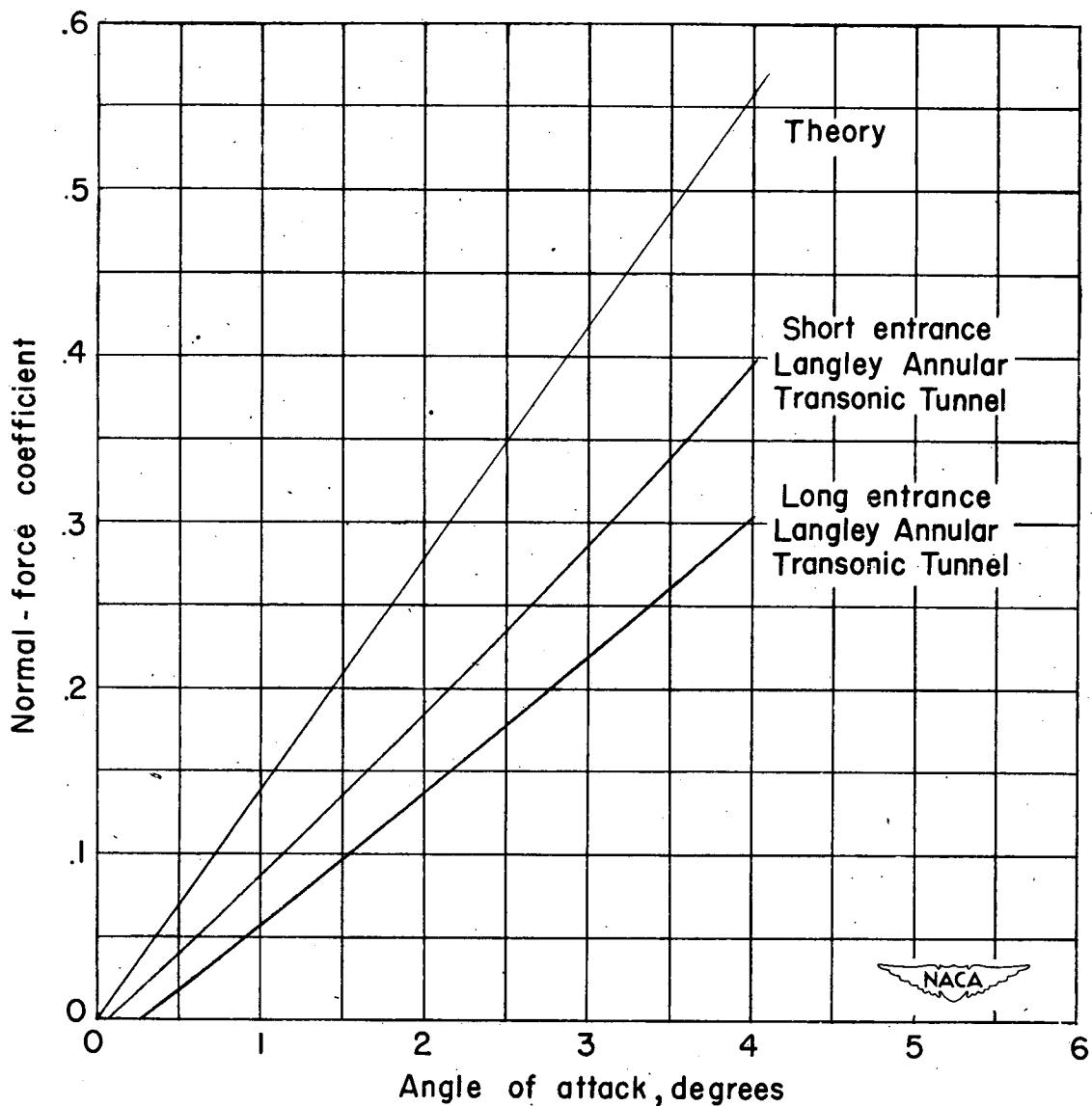
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Figure 3.- Comparison of lift-curve slopes measured in the Langley annular transonic tunnel with the theoretical lift-curve slope. NACA 66-006 airfoil; Mach number, 0.625.

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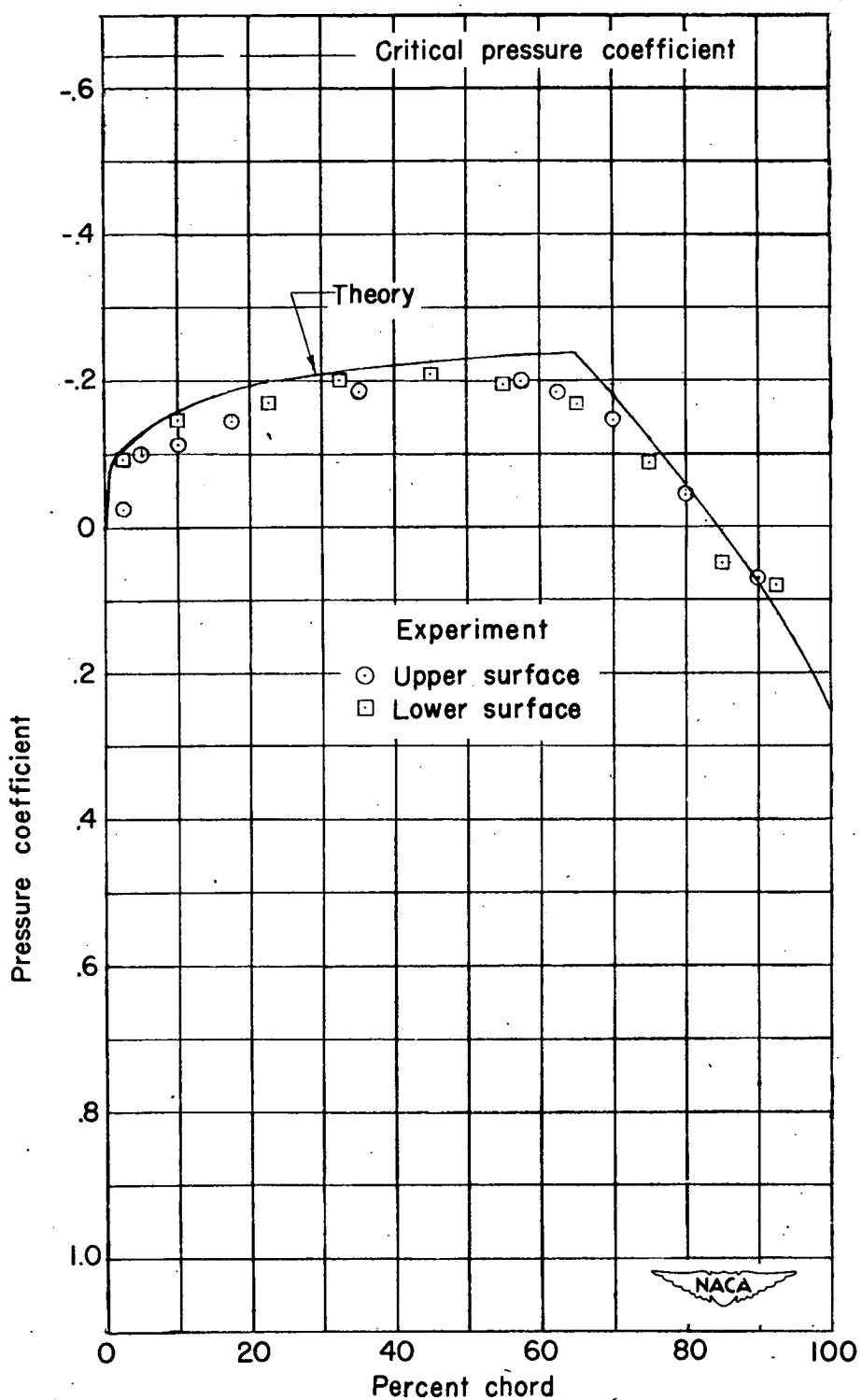


Figure 4.- Comparison of experimental and theoretical pressure distributions at  $0^{\circ}$  angle of attack. NACA 66-006 airfoil; Mach number, 0.735.

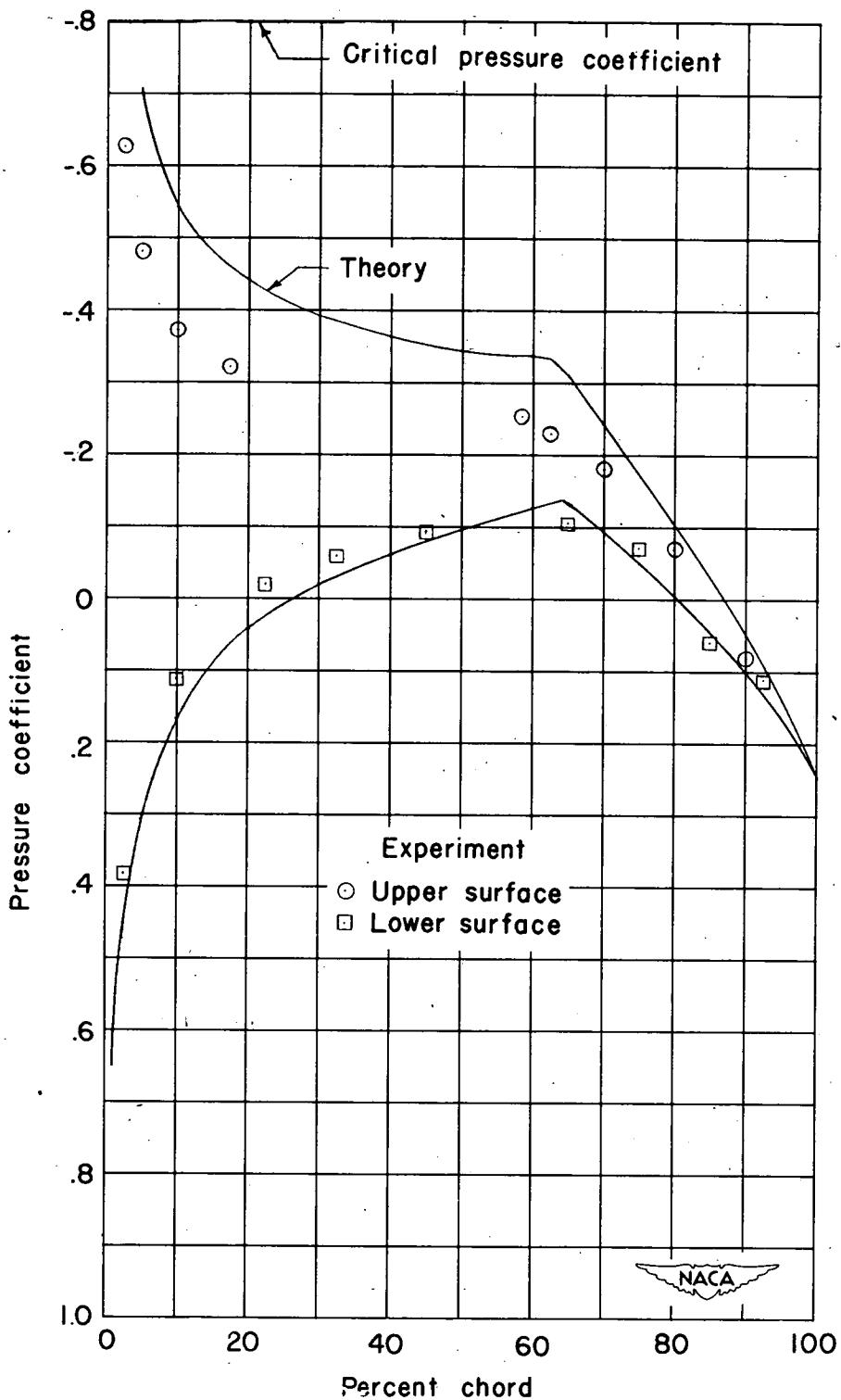
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Figure 5.- Comparison of experimental and theoretical pressure distributions at equal angles of attack. NACA 66-006 airfoil; Mach number, 0.694; angle of attack,  $2.2^\circ$ .

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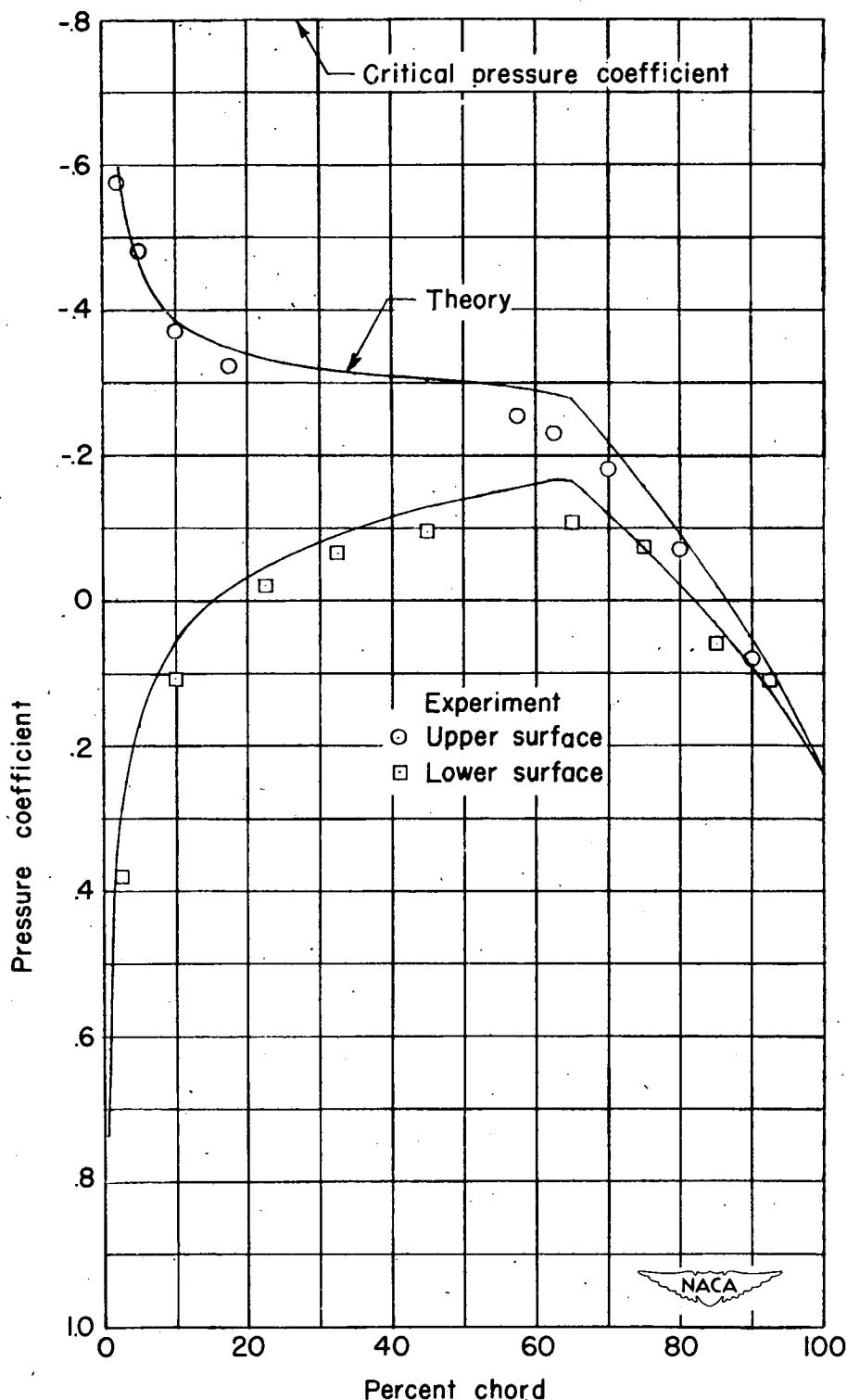


Figure 6.- Comparison of experimental and theoretical pressure distributions at equal normal-force coefficients. NACA 66-006 airfoil. Mach number, 0.694; normal-force coefficient, 0.213.

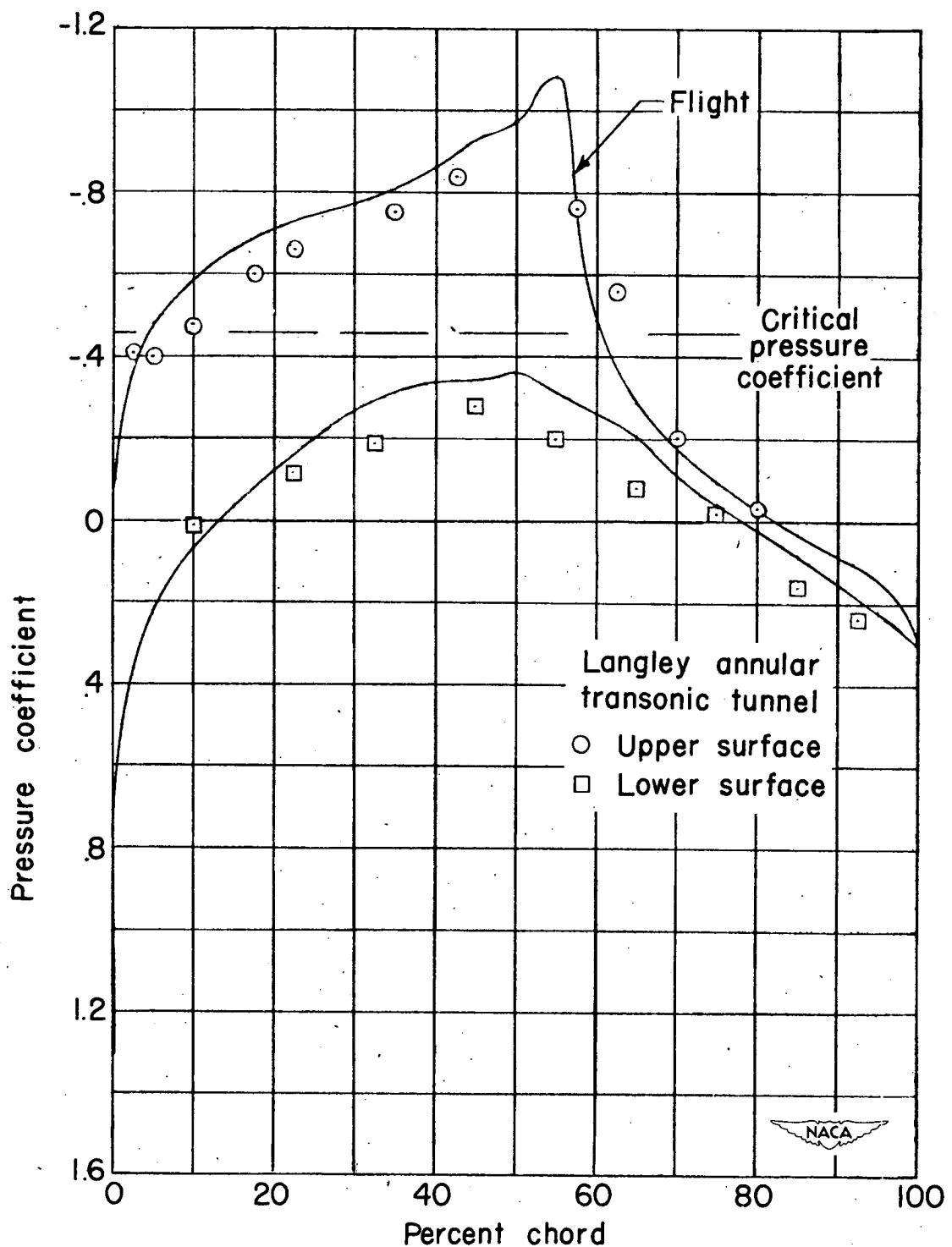


Figure 7.- Comparison of pressure distributions measured in flight and in the Langley annular transonic tunnel at equal normal-force coefficients. NACA 65-110 airfoil; Mach number, 0.79; normal-force coefficient, 0.386.

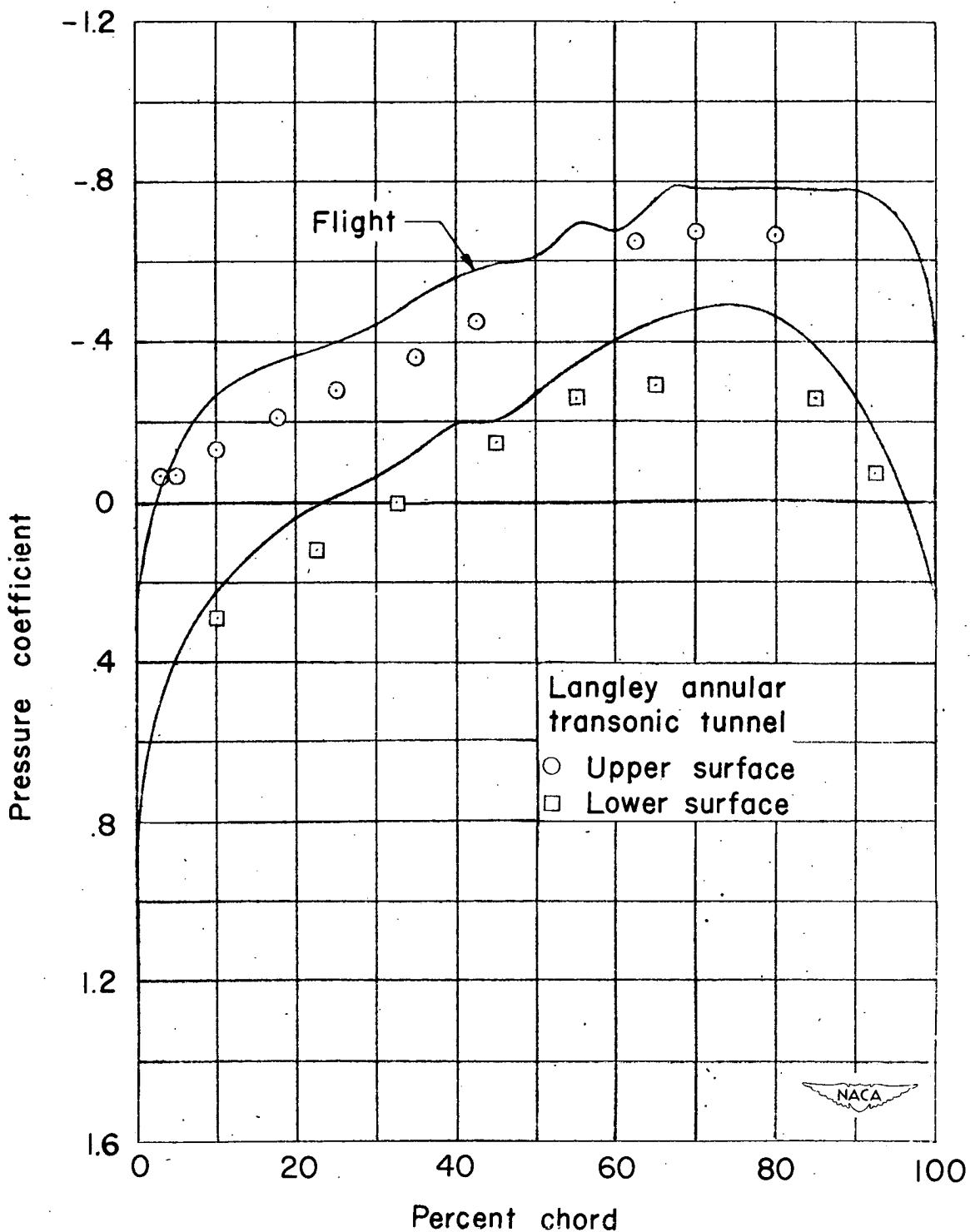


Figure 8.- Comparison of pressure distributions measured in flight and in the Langley annular transonic tunnel at equal normal-force coefficients. NACA 65-110 airfoil; Mach number, 1.00; normal-force coefficient, 0.409.

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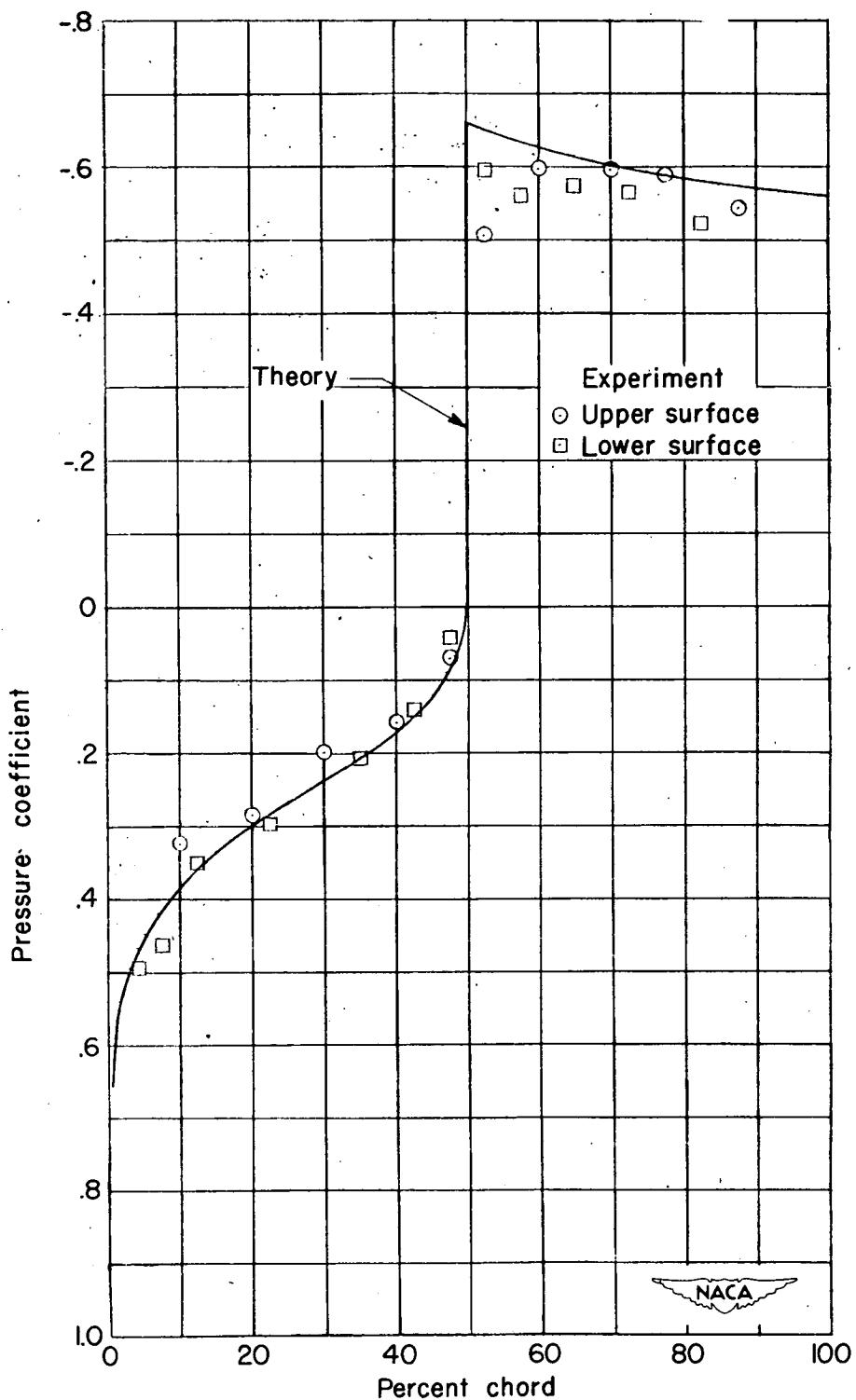


Figure 9.- Comparison of experimental and theoretical pressure distributions. Double-wedge airfoil 10 percent thick. Mach number, 1.00. Angle of attack,  $0^\circ$ .

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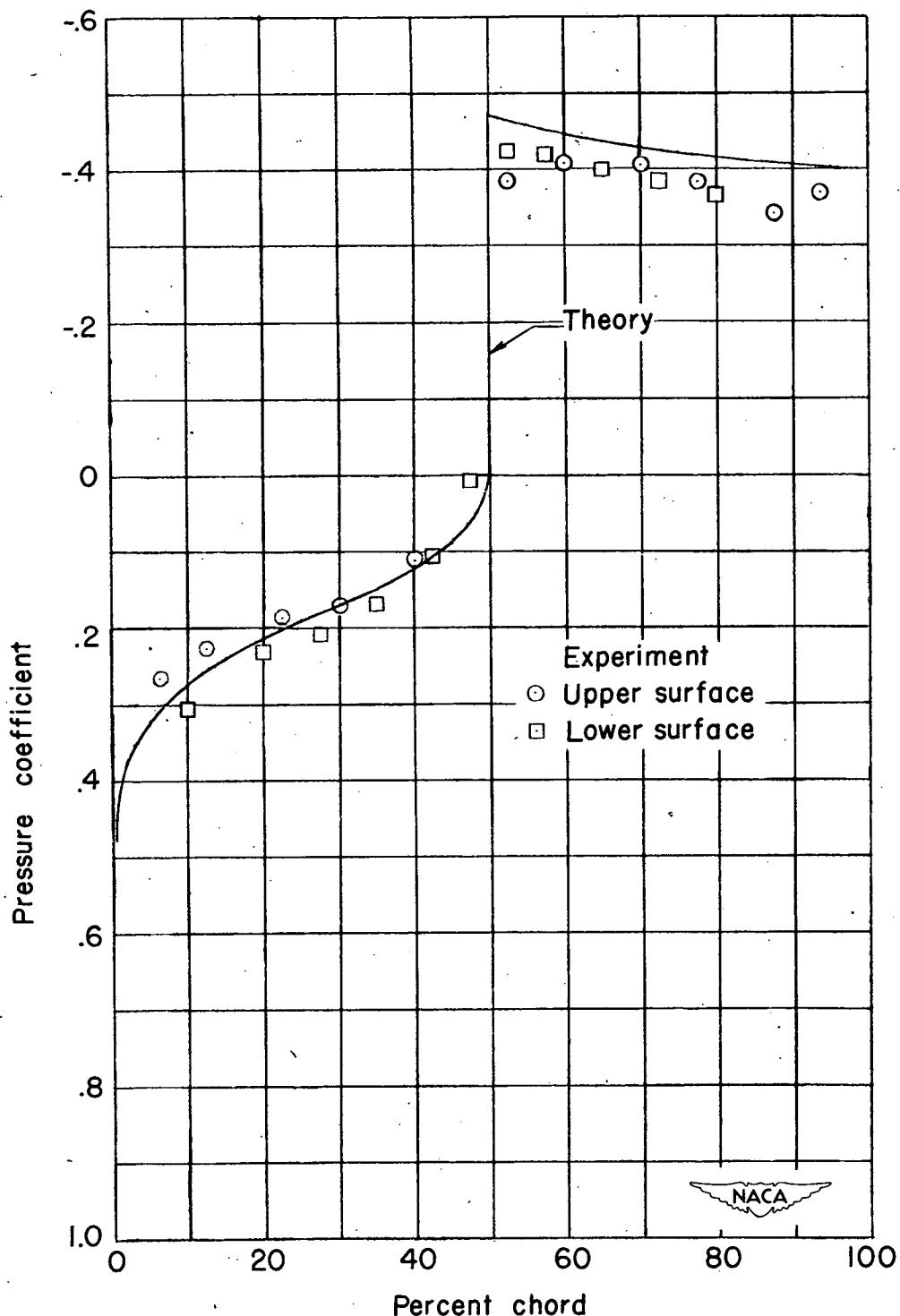


Figure 10.- Comparison of experimental and theoretical pressure distributions. Double-wedge airfoil 6 percent thick; Mach number, 1.00; angle of attack,  $0^{\circ}$ .

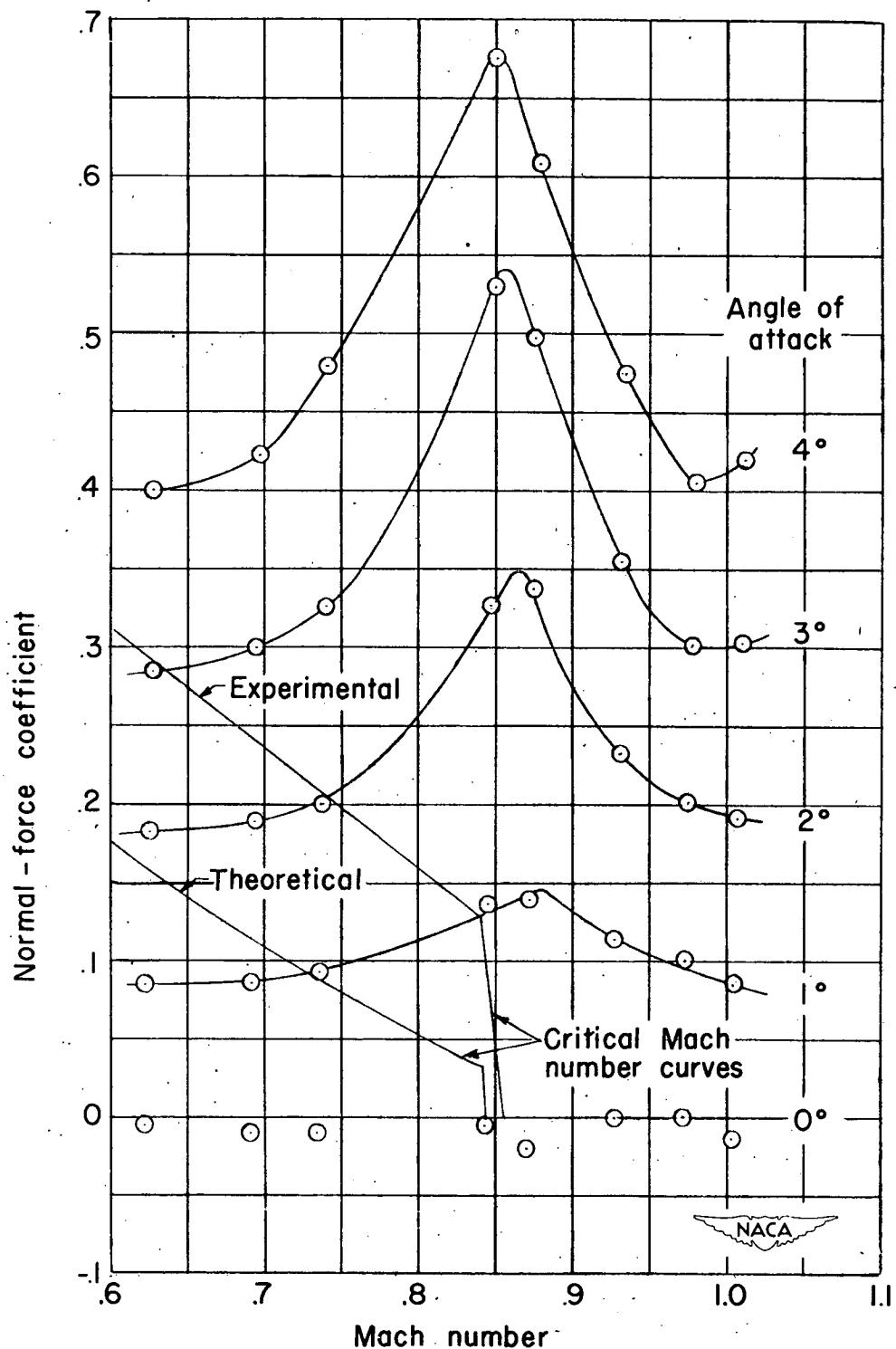
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Figure 11.- Variation of normal-force coefficient with Mach number for several angles of attack as measured in the Langley annular transonic tunnel. NACA 66-006 airfoil.

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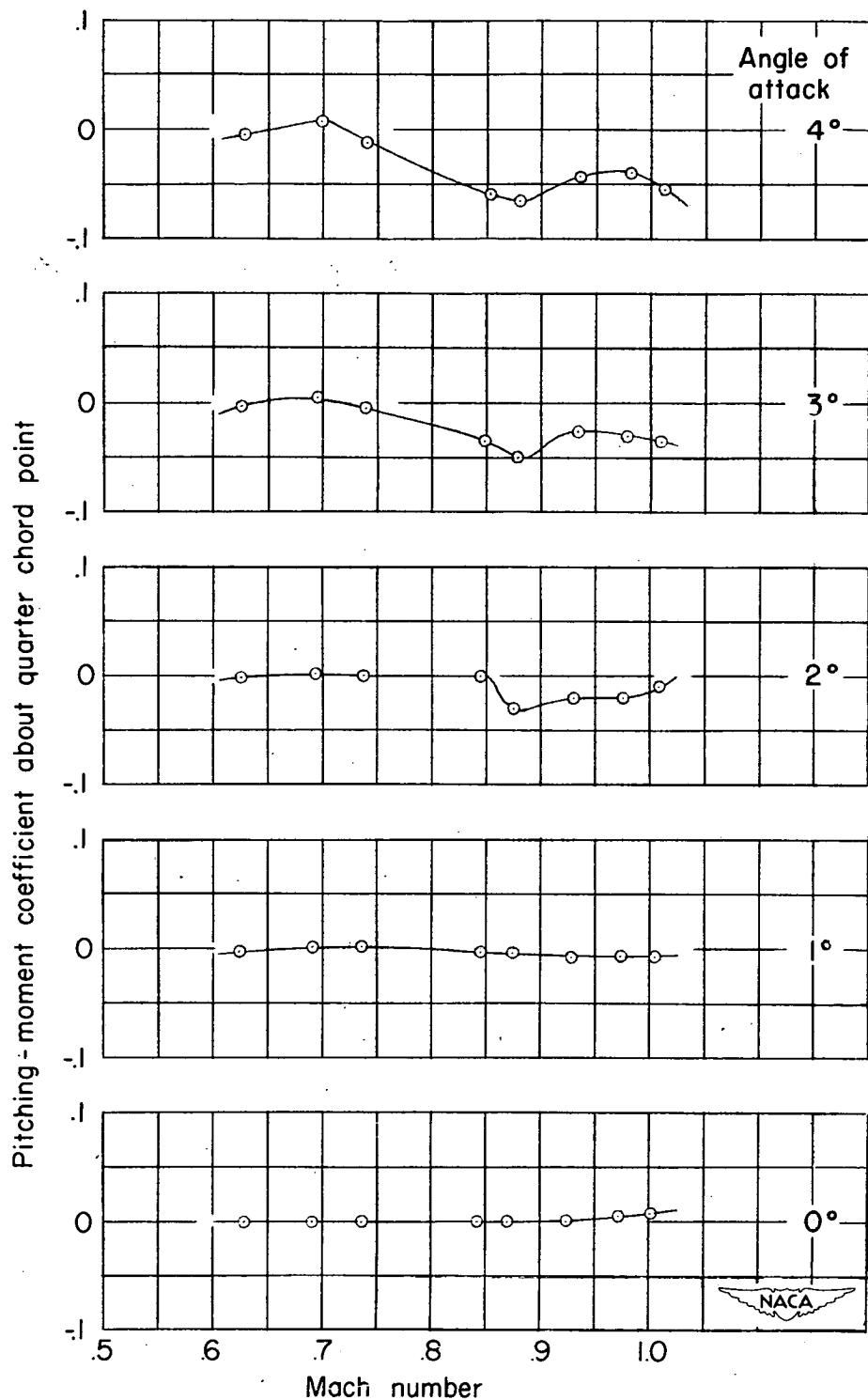


Figure 12.- Variation of pitching-moment coefficient with Mach number for several angles of attack as measured in the Langley annular transonic tunnel. NACA 66-006 airfoil.